

An Accelerated Development, Reduced Cost Approach to Lunar/Mars Exploration Using a Modular NTR-Based Space Transportation System

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**AN ACCELERATED DEVELOPMENT, REDUCED COST APPROACH
TO LUNAR/MARS EXPLORATION
USING A MODULAR NTR-BASED SPACE TRANSPORTATION SYSTEM**

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Abstract

The results of integrated systems and mission studies are presented which quantify the benefits and rationale for developing a common, modular lunar/Mars space transportation system (STS) based on nuclear thermal rocket (NTR) technology. At present, NASA's Exploration Program Office (ExPO) is considering chemical propulsion for an "early return to the Moon," and NTR propulsion for the more demanding Mars missions to follow. The time and cost to develop these multiple systems are expected to be significant. The Nuclear Propulsion Office (NPO) has examined a variety of lunar and Mars missions and heavy lift launch vehicle (HLLV) options in an effort to determine a "standardized" set of engine and stage components capable of satisfying a wide range of Space Exploration Initiative (SEI) missions. By using these components in a "building block" fashion, a variety of single and multi-engine lunar and Mars vehicles can be configured. For NASA's "First Lunar Outpost" (FLO) mission, an expendable NTR stage powered by two 50 klbf engines can deliver ~96 metric tons (t) to trans-lunar injection (TLI) conditions for an initial mass in low Earth orbit (IMLEO) of ~198 t compared to 250 t for a cryogenic chemical TLI stage. The NTR stage liquid hydrogen (LH₂) tank has a 10 m diameter, 14.5 m length, and 66 t LH₂ capacity. The NTR utilizes a UC-ZrC-graphite "composite" fuel with a specific impulse (Isp) capability of ~900 s and an engine thrust-to-weight ratio of ~4.3. By extending the size and LH₂ capacity of the lunar NTR stage to ~20 m and 96 t, respectively, a single launch Mars cargo vehicle capable of

delivering ~50 t of surface payload is possible. Three 50 klbf NTR engines and the two standardized LH₂ tank sizes developed for lunar and Mars cargo vehicle applications would be used to configure the Mars piloted vehicle for a mission as early as 2010. The paper describes the features of the "common" NTR-based Moon/Mars STS, examines performance sensitivities resulting from different "mission mode" assumptions, and quantifies potential schedule and cost benefits resulting from this modular Moon/Mars NTR vehicle approach.

Introduction

On July 20, 1989, the 20th anniversary of the Apollo 11 Moon landing, President Bush tasked the National Aeronautics and Space Administration (NASA) to undertake a Space Exploration Initiative (SEI) aimed at returning humans to the Moon "to stay" in the next century, followed by a journey to Mars using systems "space tested" in the lunar environment. Initial assessments of the space transportation system elements and infrastructures required to move humans and support equipment (e.g., habitats, supplies, and science and exploration equipment) from Earth to the surfaces of the Moon and Mars were outlined by NASA in its "90-Day Study Report"¹ issued in November, 1989. Since that time "in-house" NASA studies and contractor-funded efforts have continued to refine and improve on these initial results. In its recently released report² entitled "*America at the Threshold: America's Space Exploration Initiative*", the Synthesis Group outlined four different possible approaches or architectures for carrying out the SEI.

*Nuclear Propulsion Office

**Cost Analysis Office

†Advanced Space Analysis Office

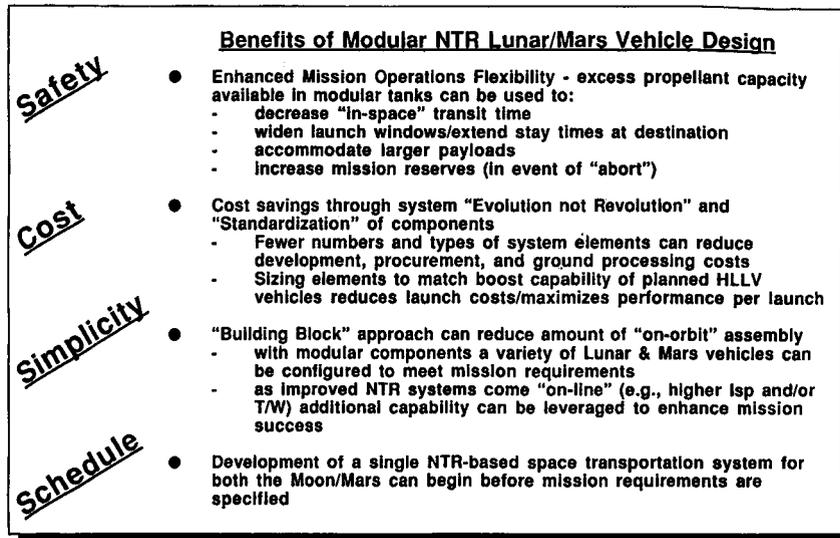


Fig. 1. Rationale for a Modular NTR Vehicle Approach

The Synthesis Group also specified several important technical strategies common to its four architectures that affect space transportation systems design. These included use of (1) a heavy lift launch vehicle to limit on-orbit assembly; (2) a split mission strategy (where cargo and crew fly on separate missions); (3) pre-deployed and verified "turn-key" habitats; (4) chemical and nuclear thermal propulsion for lunar and Mars missions, respectively; (5) direct entry of returning crews to Earth's surface; (6) lunar missions as a "testbed" for Mars, and (7) to the extent possible, common systems for lunar and Mars missions.

At present, the ExPO is baselining chemical propulsion for its FLO mission in the 2000 to 2003 timeframe, and NTR propulsion for the more demanding Mars missions beginning in the 2007 to 2010 timeframe. The selection of NTR propulsion is in keeping with the Synthesis Group report which recommended the NTR as the "only prudent propulsion system for Mars transit."³ Because the time and cost to develop two separate transportation systems for SEI could be substantial, the Nuclear Propulsion Office (NPO) has been examining^{4,5,6} the rationale and benefits of developing a "fully reusable" NTR-based lunar STS and then evolving it to Mars mission

applications through the use of modular engine/stage components. In this present work, a common, modular NTR-based STS is proposed which uses "standardized" engine and stage components in a "building block" fashion to configure a wide variety of single and multi-engine lunar and Mars vehicles.

The ability to develop a NTR lunar transfer stage in time to support NASA's proposed FLO mission was shown to be feasible in a recently completed study⁷ performed by the NPO with support from the Department of Energy (DOE) and industry contractors. The NPO's projected schedule for NTR development calls for ground testing of a "prototype" flight engine in 2003, with qualification flights to the Moon initiated shortly afterwards. In addition to obtaining valuable operational experience, the use of NTR propulsion for lunar missions would enhance performance and allow NASA to make a significant down payment during its initial lunar program on key components of the modular STS needed for the subsequent Mars mission. The modular approach has a number of attractive features (see Fig. 1) which include enhanced mission flexibility and safety, simplified vehicle design and assembly, and reduced development/procurement costs through

standardization of the fewest number of components. An accelerated, reduced cost approach to overall lunar/Mars exploration is therefore expected.

This paper describes the results of integrated systems and mission studies used to determine the engine and stage characteristics which are best suited for a modular STS approach and which are also compatible, from a mass and volume standpoint, with a range of possible HLLV options. The paper also includes schedule and cost information consistent with a development program supporting full engine "ground tests" in 2003 and 2004, initial lunar applications in 2005 and 2006, and then Mars cargo and piloted flights in 2007 and 2010, respectively. The paper first describes the operating principles and "state-of-the-art" characteristics of a NTR concept derived from technology "demonstrated" in the Rover/NERVA nuclear rocket programs. The lunar and Mars mission scenarios currently envisioned by NASA are then reviewed. Mission and transportation system ground rules and assumptions used in this study are presented next along with descriptions and mass summaries of several modular lunar and Mars space transfer vehicles. The programmatic and costs required to support development of the modular STS follows next. Finally, a summary of results and the conclusions reached in the study are presented.

NTR Systems Description and Characteristics

The nuclear thermal rocket represents the next major evolutionary step in propulsion technology and is expected to be an important complement to chemical propulsion for NASA's SEI missions. Conceptually, NTR systems are relatively simple (see Fig. 2). They function by raising hydrogen propellant to high pressure in a turbopump assembly, passing it through a high power reactor where it is heated to high temperatures, and then exhausting it through a nozzle at high speeds to generate thrust. Because a fission reactor, rather than chemical reactants, provides the heat source, the NTR can use low molecular weight liquid hydrogen as both the reactor coolant and propellant and achieve specific impulse values nearly twice that of conventional liquid oxygen/liquid hydrogen (LOX/LH₂) fueled chemical rockets at comparable exhaust temperatures.

In the "expander cycle" engine shown in Fig. 2, the turbine drive gas is routed to twin turbopumps (used for redundancy and improved system reliability) and then through the reactor core allowing the entire propellant flow to be heated to design conditions. Hydrogen flowing from the pumps would be split with a portion being used to cool the nozzle, reflector, control rods and internal dome

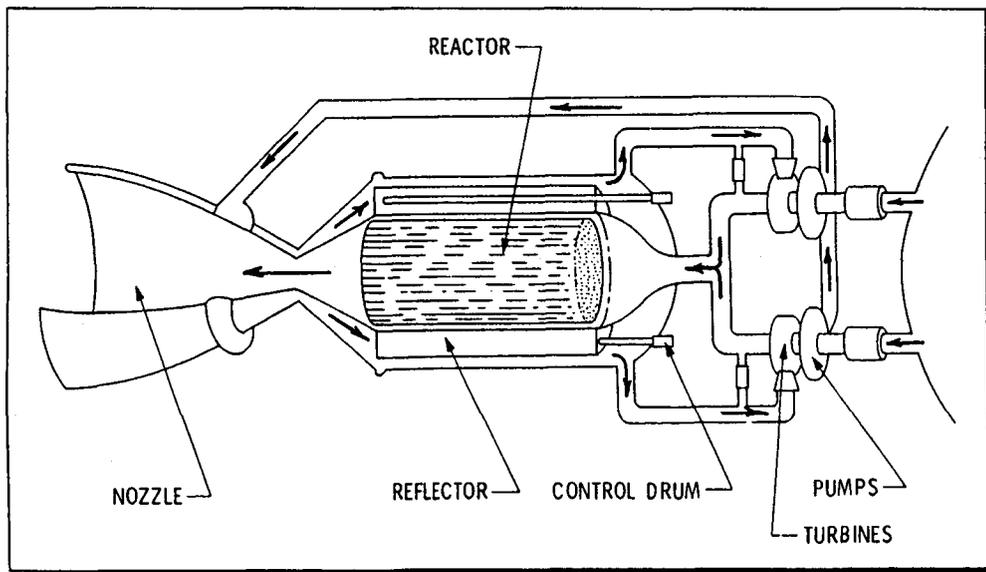


Fig. 2. Schematic of Dual Turbopump Expander Cycle NTR

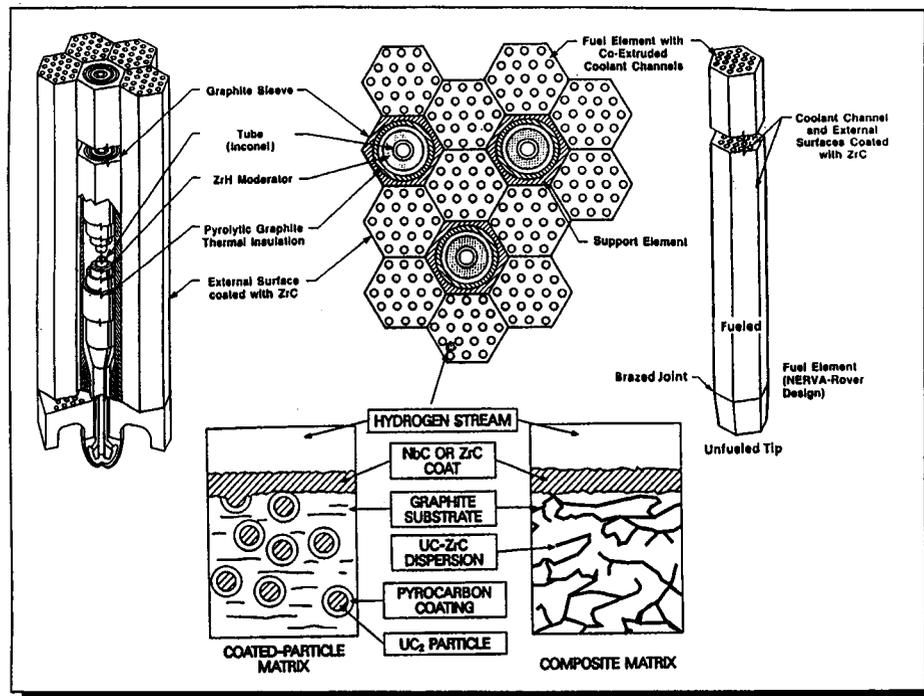


Fig. 3. Rover/NERVA Fuel Element Configuration

shield, and the remainder going to the core support tie tubes (not shown in Fig. 2) for cooling and providing the necessary turbine drive power.

Rover/NERVA Technology Overview

The feasibility of a hydrogen-cooled, graphite moderated NTR was demonstrated by the Rover nuclear rocket program⁸ begun at Los Alamos in 1955. The promising early results from this effort led to the formation in 1960 of a joint program between NASA and the Atomic Energy Commission (AEC) to develop a Nuclear Engine for Rocket Vehicle Application (NERVA)⁹. From 1955 until the program was stopped in 1973, a total of twenty reactors were designed, built and tested at a cost of ~\$1.4 billion. Escalated to 1992 dollars, Rover/NERVA technology represents an investment of ~\$10 billion.

At the heart of the NERVA reactor design is a 52" long hexagonally-shaped fuel element (0.75" across the flats), which is capable of producing approximately 0.9 to 1.2 megawatts of thermal power (MWt) (see Fig. 3). Each fuel element has 19 axial coolant channels, which along with the outer

element surfaces, are coated with zirconium carbide (ZrC) to reduce hydrogen/graphite reactions. A "2-pass" regeneratively-cooled, tie-tube assembly supports from 3 to 6 fuel elements forming a fuel bundle (shown in Fig.3). Specifying the engine thrust level, hydrogen exhaust temperature (or equivalent Isp), and the fuel element power density determines the reactor power output and sets the core diameter and number of fuel bundles required in the engine. For lower thrust engines, criticality can be achieved with reduced core diameters and acceptable thrust-to-weight ratios by augmenting the moderating capability of the graphite core with additional zirconium hydride (ZrH) neutron moderator. The ZrH is contained in the tie-tube support elements which are increased in number for lower thrust engines by decreasing the fuel-to-support element ratio (from ~6 to 1 for engine thrust levels of ~50 kbf or greater, down to ~3 to 1 for a 25 kbf-class engine).

Two of the fuel forms tested⁸ during the Rover/NERVA programs are also shown in Fig. 3. The majority of experimental testing was performed using "graphite" fuel. It consisted of pyrocarbon coated uranium carbide (UC₂) fuel

particles which were dispersed in a graphite substrate (see Fig. 3). This fuel was operated at hydrogen exhaust temperatures as high as 2550 K. The second fuel form was a "composite" fuel which consisted of a UC-ZrC dispersion in the graphite substrate. Although the composite fuel received only limited nuclear testing in the Nuclear Furnace (NF-1)⁸, it also underwent extensive electrical furnace testing¹⁰ (~10 hours at 2750 K with 64 temperature cycles) which demonstrated the potential to provide hydrogen exhaust temperatures and equivalent Isp values of ~2700 K and 900 s, respectively. Because of its growth and performance potential the composite fuel form was selected as the reference fuel form in this study.

Engine Performance Characteristics and Sizing

As part of the earlier "Fast Track" NTR Study⁷ conducted by NPO, DOE and industry, point designs were developed for 25, 50 and 75 klb-class engines utilizing both graphite and composite fuel. Performance projections for these systems are shown in Table 1, and indicate the potential for substantial improvements in both Isp and engine thrust-to-weight ratio over the 1972 NERVA reference engine. Modest increases in chamber temperature, pressure, and individual fuel element

power output (from ~0.9 MWt to ~1.2 MWt) were assumed along with a nozzle area expansion ratio of 200 to 1 and a 110% length optimum contour Rao nozzle. An expander cycle was also baselined in this study with turbine drive gas provided by the reactor tie-tube support elements. Finally, dual centrifugal turbopumps and an internal radiation shield (comprised of boron-carbide aluminum-titanium hydride (BATH) and lead) were included in the engine weight estimates to provide redundancy, and improve engine reliability and safety. The relative size of the 25, 50 and 75 klb-class, composite fuel engines is shown in Fig. 4.

Proposed Lunar/Mars Mission Scenarios

A large number of options for lunar¹¹ and Mars¹² exploration were studied and proposed by NASA during the 1960's as possible follow-on activities to the "post-Apollo" program. Since 1987, the Office of Exploration at NASA Headquarters and the NASA field centers have been conducting studies^{13,14,15} aimed at determining the technologies, systems and infrastructure needed to support future lunar and Mars exploration initiatives. An overview of the current lunar and Mars mission scenarios being considered by NASA is provided in the sections which follow.

Table 1. Characteristics of NERVA-Type Engines

Parameters	72 NERVA*		"State-of-the-art" NERVA Derivatives*				
	Hot Bleed/Expander		Expander				
Fuel Form	Graphite		Graphite		Composite		
Thrust (klbf)	75	25	50	75	25	50	75
Chamber Temperature (K)	2350		2550		2700		
Chamber Pressure (psia)	450		785		785		
Nozzle Expansion Ratio	100:1		200:1		200:1		
Specific Impulse (sec)	825/845		870		900		
Engine Mass (kg)**	11250	3727	4762	6205	3883	5237	6823
Engine Thrust/Weight **	3.0	3.0	4.8	5.5	2.9	4.3	5.0

* Engine masses contain dual turbopump capability for redundancy.
 ** Includes internal shield but no external disk shield mass.

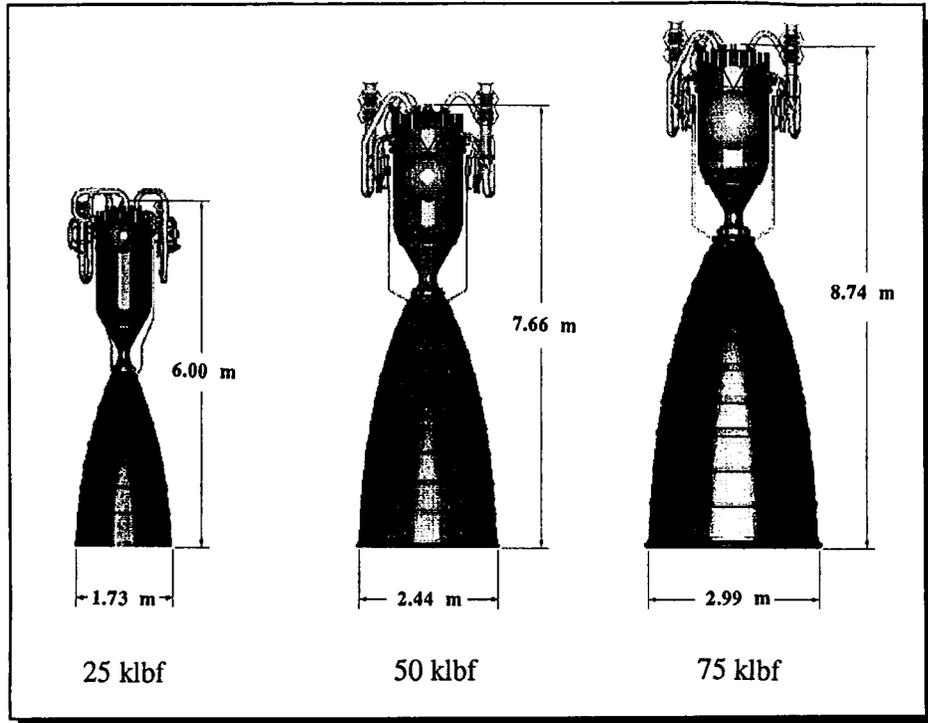


Fig. 4. Relative Size of Dual Turbopump NTR Engines

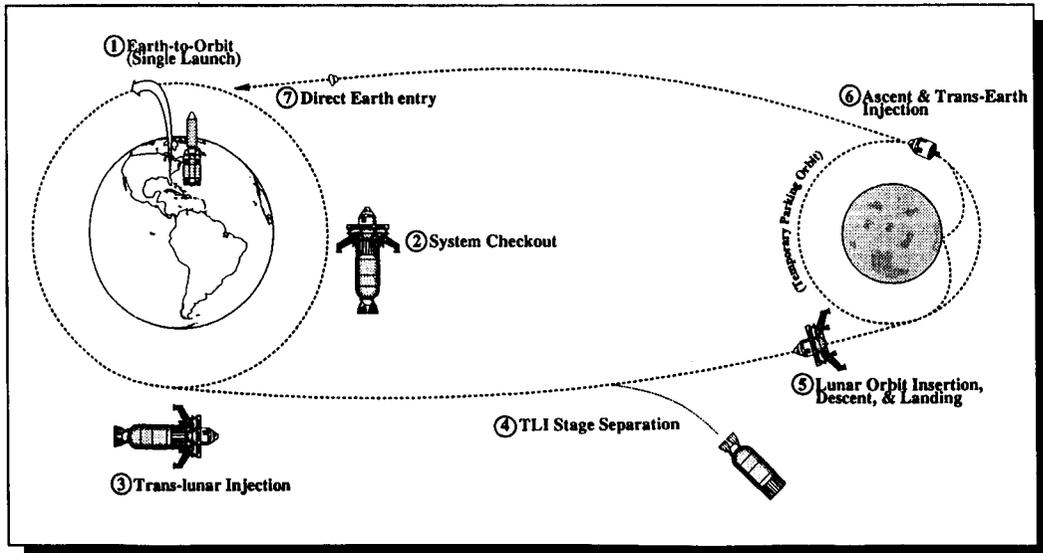


Fig. 5 "First Lunar Outpost" Piloted Mission Scenario

First Lunar Outpost Mission

NASA has spent considerable time assessing the human operations and surface support requirements needed to return humans to the Moon at levels ranging from short expeditionary landings to a centralized base¹⁶ supporting a substantial permanent human presence. Following its review of the Synthesis Group architectures, the ExPO adopted a "lunar campsite" strategy¹⁷ for the FLO mission. Designed to provide facilities to support a crew of four for 45 Earth-days (i.e., a lunar day, night, day cycle), FLO consists of a pre-integrated, reusable habitat module delivered intact on a cargo lander. The outpost would be autonomously landed and its operational functions verified prior to crew arrival on a separate piloted flight. This predeployment of surface infrastructure via the split cargo and piloted mission approach is expected to improve overall mission success and reduce the amount of EVA required by the crew to prepare the outpost for initial occupancy.

The FLO mission assumes separate cargo and piloted missions each of which are launched on a single 250 t-class HLLV. The piloted mission utilizes a "lunar direct" mode which provides

"global access" to the Moon and an "anytime abort" capability for the crew. Key phases of the piloted mission scenario are depicted in Fig. 5. The mission begins with the launch of a single 250 t HLLV, which delivers the piloted vehicle and TLI stage to a circular 100 nautical mile (185 km) Earth staging orbit. Here the vehicle systems are checked out and verified prior to Earth departure. The expendable TLI stage is then fired, placing the piloted vehicle on a 4-day trajectory to the Moon. After transfer to the Moon is complete, the lunar lander is used to propulsively capture the piloted vehicle into a temporary 100 km parking orbit. Pausing here allows time for navigational updates and phasing alignment over the desired landing site prior to final descent to the lunar surface. When the surface mission is completed, the crew reenters the return stage and ascends to its earlier parking orbit prior to initiating trans-Earth injection (TEI). Nearing Earth, the crew module separates from the return stage and performs a direct Earth entry, while the return stage is expended in cislunar space via an Earth fly-by.

An NTR-based FLO scenario analogous to the chemical mission is illustrated in Fig.6. The NTR stage with its two 50 kbf thrust engines is assumed to depart from the same 185 km as the chemical system. Higher LEO starting altitudes, on the order

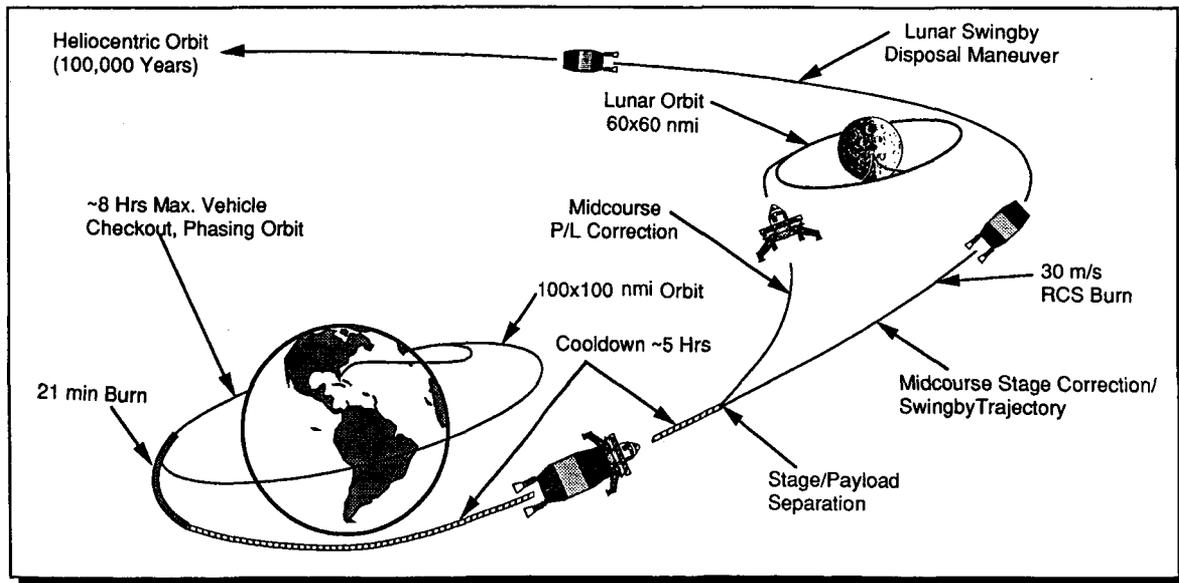


Fig. 6. NTR-Based FLO Mission Scenario

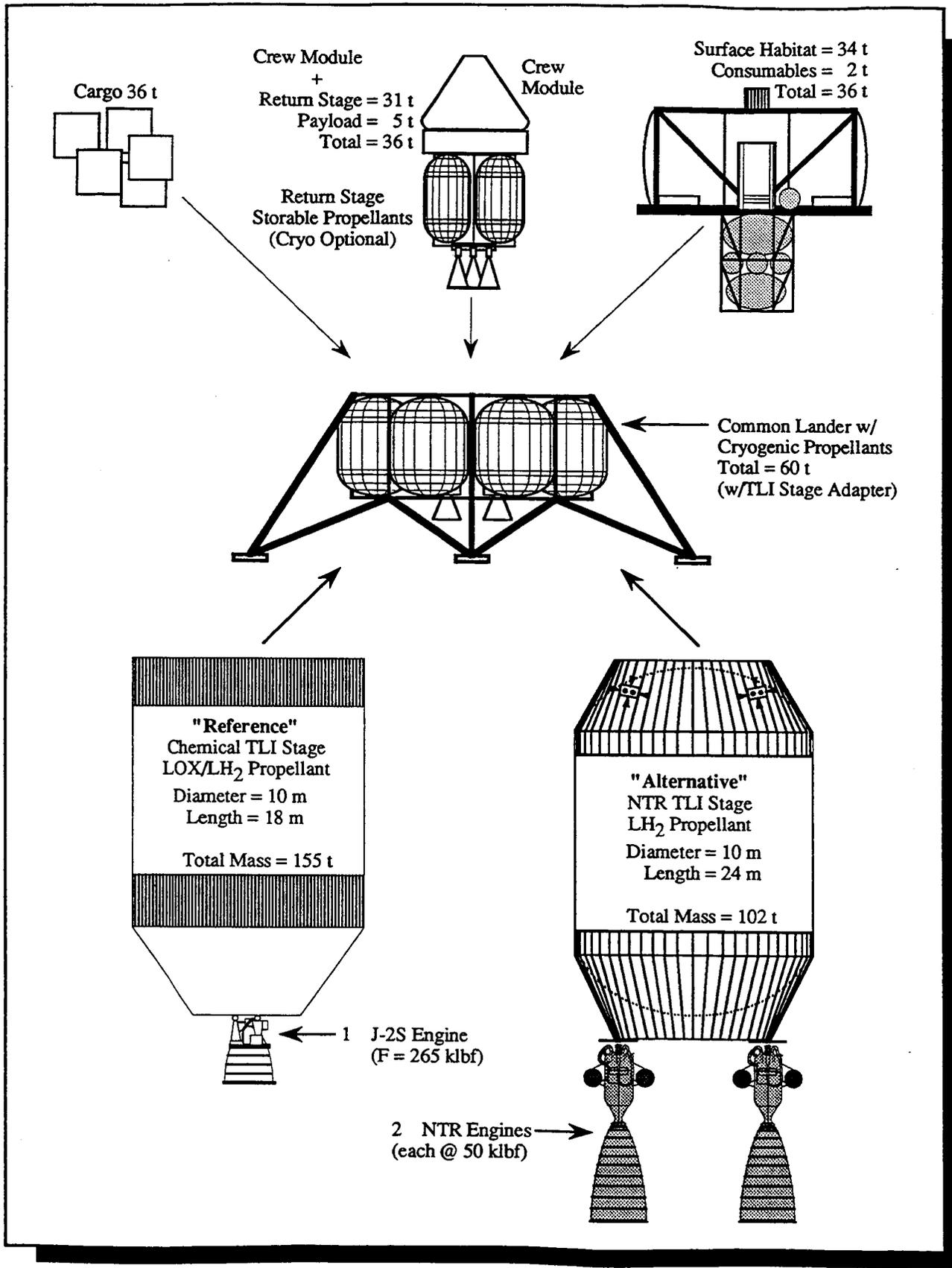


Fig. 7. FLO Space Transportation System Elements

of 260 nautical miles (~480 km), can readily be achieved with the reference 250 t HLLV because of the substantially lighter mass of the NTR stage compared to that of the chemical system.

Following a TLI burn lasting ~21 minutes and an appropriate cooldown period, the piloted vehicle and NTR stage separate with the piloted vehicle continuing on its nominal mission. After separation, the NTR stage executes a retargeting maneuver with its RCS system to perform a "trailing edge" lunar swingby. The resulting lunar gravity assist is used to deliver the "spent" NTR stage to a long-lived (~10⁵ year) heliocentric orbit with minimal risk of Earth reencounter.

The main elements of the FLO space transportation system are shown in Fig. 7. They consist of a TLI stage, a common lunar lander, an Earth return stage, and a crew module, all of which are expended during the course of the mission. In the "cargo only" mode, the return stage and crew module would be replaced by an equivalent amount of cargo which could include such items as surface habitats, crew consumables, rovers and science equipment. The total mass of the common lander with its cryogenic propellant load, payload and TLI stage adaptor is ~96 t.

The reference chemical TLI stage uses a single J-2S engine which has a Isp and thrust level of 436 s and 265 klbf, respectively. The stage contains ~133.5 t of LOX/LH₂ propellant and has an inert mass of ~21.5 t. The alternative NTR stage uses two 50 klbf engines which operate at a Isp = 900 s and provide a total thrust of 100 klbf. Although the stage is ~6 m longer than the chemical system, it is ~52 t lighter than its chemical counterpart. The propellant and stage inert weights are ~66 t and 36 t, respectively.

Mars Mission Scenarios

Over the past several years, NASA has been examining the advantages and disadvantages of various trajectory classes, mission opportunities, and propulsion system options for its piloted missions to Mars.^{18,19} From these and other studies² the NTR has emerged as the leading candidate technology for primary space propulsion. This lead

role is attributed both to its maturity (a large experimental database was accumulated during the Rover/NERVA programs) and to its high Isp which enables the NTR to leverage a given propellant loading to reduce the total "in-space" transit time.

In FY'89 and '90, NASA's reference Mars mission was an "all-up", 434 day, 2016 opposition-class mission with a 30-day surface stay and an inbound Venus swingby. "All-up" refers to an operational mode in which all of the payload and propellant required for the complete Mars mission is carried on a single vehicle. Prior to FY'89, NASA spent several years examining the benefits of splitting the "all-up" Mars mission into two parts -- a cargo mission and a piloted mission. In this so-called "split cargo/piloted sprint" mission mode, cargo would first be transported to Mars by a cargo vehicle(s) taking a slow, minimum propellant, low energy trajectory to Mars. The piloted vehicle would travel to Mars on a faster, higher energy trajectory after receiving confirmation that the cargo vehicle(s) had arrived safely in Mars orbit. By employing a "fast transit time" strategy, it is felt that crew health hazards resulting from long term exposure to weightlessness and space radiation can be minimized.

Three basic split/sprint mission modes are available for consideration.²⁰ In the "all-up" mode, the piloted transfer vehicle (PTV) carries its own Mars excursion vehicle (MEV) and all of the propellant required for the fast-return transit to Earth. The corresponding cargo transfer vehicle (CTV) carries only an autonomous lander outfitted with the necessary supplies to support the surface mission. In the "No MEV" mode, the PTV carries only its return propellant and lands on Mars with a MEV carried on the CTV. A rendezvous in Mars orbit is therefore required between the PTV and CTV. The third option, the "No MEV/No TEI Propellant" mode, uses CTVs to pre-deploy all cargo including Earth-return propellant at Mars. The TEI propellant can be transported either in a "tanker" CTV or in a separate "return stage". Both techniques still require a Mars orbit rendezvous between the PTV and CTV, but the later option would eliminate the need for propellant transfer.

The ExPO is presently assessing the requirements for supporting a piloted mission to Mars as early as 2010 using the "all-up" split/sprint mission approach at its baseline. Key phases of the piloted mission profile are illustrated in Fig. 8. The piloted mission is preceded by three individual cargo missions which depart Earth orbit in September, 2007 and arrive at Mars ~343 days later. Each NTR cargo vehicle is launched on a single 250 t HLLV and transports a 63 t MEV capable of landing ~50 t of payload on the Mars surface. After completing its assembly and checkout in low Earth orbit, the piloted vehicle leaves Earth in November, 2009. It arrives at Mars ~6 months later using a "fast conjunction-class" trajectory,^{18,19} which maximizes the exploration time at Mars while reducing the total in-space transit time to under a year. After a 530 day stay at Mars, the crew returns in the ascent portion of the MEV to the orbiting piloted vehicle and begins its preparation for a 5 month journey back to Earth. The total duration for the piloted mission is just over 880 days. Crew return to Earth is via an Earth crew capsule vehicle (ECCV) similar to that used in the Apollo program, while the NTR vehicle and habitat are jettisoned for disposal into deep space.

Mission/Transportation System Ground Rules and Assumptions

The ground rules and assumptions used to determine the characteristics of the modular engine and stage components are representative of those currently being considered by ExPO. Tables 2, 3 and 4 summarize the information used in assessing the FLO and Mars missions. Included are details on payload masses (e.g., MEV, crew habitat, ECCV, etc.), parking orbits, primary and auxiliary propulsion, tankage, and mission velocity change (ΔV) requirements. In addition to the ΔV s for the primary propulsive maneuvers performed by the NTR system, the NTR vehicles also execute mid-course and secondary maneuvers using a storable, bipropellant reaction control system (RCS). For the FLO, this includes a retargeting maneuver by the TLI stage to set up a lunar gravity assist for stage disposal. For the Mars cargo and piloted missions, Mars orbital operation maneuvers on the order of 100 m/s are provided for by the RCS system. Gravity losses are also taken into account for all TLI and TMI maneuvers. For the FLO and Mars cargo missions a "single burn" Earth departure is used exclusively,

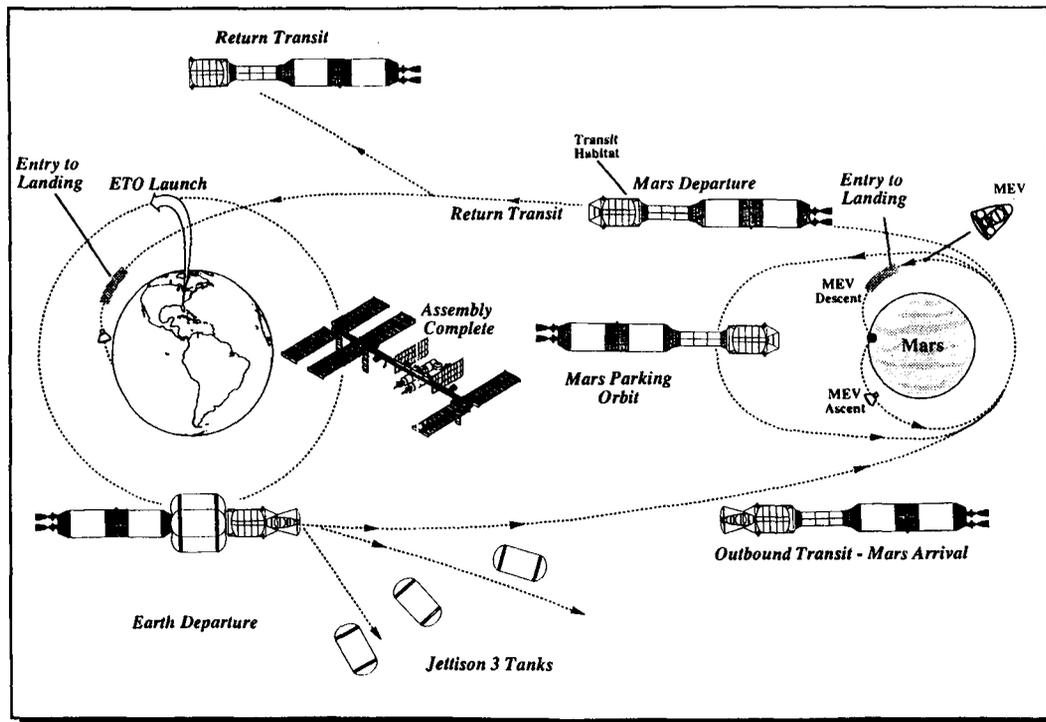


Fig. 8. NTR Piloted Vehicle Mission Profile

Table 2. FLO Mission Ground Rules and Assumptions

<u>"One Burn" Lunar Scenario</u>		
• <u>TLI Payload</u>	96 t (piloted vehicle & TLI stage adaptor)	
• <u>TLI Maneuver</u>	ΔV	3200 m/s + gravity losses
	Initial orbit	100 n. mi. circular LEO (185 km)
• <u>NTR System</u>	Propellant	Cryogenic hydrogen
	Isp	900 sec (composite) / 870 sec (graphite)
	External Shield Mass	\approx 60 kg/klbf thrust
	Burn Duration	\leq 30 minutes
	Flight Performance Reserve	1% of usable propellant
	Cooldown (effective)	3% of usable propellant
	Residual	1.5% of total tank capacity
• <u>RCS System</u>	Propellant	Hydrazine
	Isp	237 sec
	TLI burnout ΔV	60 m/s (30 m/s for trailing edge lunar flyby)
• <u>Tankage</u>	Material	2219-T87 Al
	Diameter	10 meters
	Geometry	Cylindrical tank with $\sqrt{2}/2$ domes
	Insulation	2" MLI + micrometeoroid shield (3.97 kg/m ²)
	Boiloff	12.40 kg / day
• <u>Contingency</u>	Engine & external shields	15%
	All other dry masses	10%

while for the heavier Mars piloted vehicle a "triple perigee burn" scenario was adopted. With the perigee propulsion technique, propulsive energy can be imparted to the spacecraft more effectively. This reduces the gravity losses associated with a finite burn duration and a reduced thrust-to-initial vehicle weight ratio.

Composite fuel is used almost exclusively in this study, although performance using graphite fuel is also shown for FLO. Biological external disk shields are baselined on all piloted missions. The shield weights were scaled with thrust/power level and calibrated with earlier NASA contractor studies^{21,22} of lunar NTR stages conducted in the 1960's and the early 1970's. Allowances for flight performance reserve, post-burn reactor cool down and tank trapped propellant residuals were also accounted for in estimating the total propellant requirements for the mission.

Aluminum alloy 2219-T87 ($F_{tu}=62$ ksi, $p=0.102$ lbm/in³=2821 kg/m³) was utilized for structure and LH₂ propellant tank construction. This selection is

due to its favorable properties at cryogenic temperatures and its extensive use in cryogenic tank construction. It has a relatively high strength-to-density ratio, good toughness and availability, is weldable and low in cost. Alloy 2219-T87 plate is also presently used for the LOX/LH₂ external tank on NASA's Space Shuttle. Tank thicknesses were calculated assuming a maximum internal pressure of 35 psi (241.3 kPa) and included hydrostatic loads using a "4-g" load factor along with a safety factor of 1.5. A 2.5 percent ullage was also assumed.

A two inch helium-purged, multilayer insulation (MLI) system (at 50 layers per inch) was assumed for thermal protection on the expendable FLO TLI stage. This insulation thickness exceeds the requirements for this short duration (\leq 8 hrs), "one-burn" mission, as well as, the "ground hold" thermal protection requirements for "wet-launched" LH₂ tanks (a minimum of 1.5 inches of helium-purged insulation).²³ Its use on FLO ensures extra margin and also provides the capacity for longer duration lunar missions (~30-180 days in lunar orbit), as well as, "1-way" Mars cargo missions. The 2" MLI system

Table 3. Mars Mission Ground Rules and Assumptions

<u>Mission</u>			
• Payload Outbound	Cargo 3 X 63.0 t	Piloted 75.0 t	Mars Excursion Vehicles
	-	59.1 t	Crew Habitat
• Payload Return	-	59.1 t	Crew Habitat
	-	6.8 t	ECCV
	-	1.0 t	Mars Return Samples
• Parking Orbits	407 km	407 km	Earth Departure (circular)
	250 km x 1 sol		Mars Arrival/Departure
• Perigee Burns	1	3	Earth Departure
• Crew Size	-	6	
<u>Propulsion</u>			
• NTR System			
Propellant		Cryogenic Hydrogen	
Isp		900 sec (composite)	
External Shield Mass		≈ 60 kg/klbf thrust	
Burn Duration		≤ 30 minutes	
Flight Performance Reserve		1% of usable propellant	
Cool down (effective)		3% of usable propellant	
Residual		1.5% of total tank capacity	
• RCS System			
Propellant		N ₂ O ₄ /MMH	
Isp		320 sec	
<u>Structure</u>			
• Tankage			
Material		2219-T87 Al	
Diameter		10 m	
Geometry		Cylindrical tank with $\sqrt{2}/2$ domes	
• Insulation			
Cargo		2" MLI + micro shield (3.97 kg/m ²)	
Piloted			
"Core Stage" & "In-line" tanks		4" MLI + micro shield+VCS (7.53 kg/m ²)	
TMI "Drop" Tanks		2" MLI + micro shield (3.97 kg/m ²)	
• Contingency			
Engine & External Shield		15%	
All other dry masses		10%	
<u>Boiloff</u>			
• Cargo Vehicle		0.769 kg/m ² /month	
• Piloted Vehicle			
"Core Stage" & "In-line" tanks		0.375 kg/m ² /month	
TMI "Drop" Tanks		0.769 kg/m ² /month	
<u>Miscellaneous</u>			
• Gravity losses modelled for Earth departure only			

Table 4. Mars Cargo and Piloted Mission ΔV Budgets

Mission Type	Launch Date	Total Mission Time (days)	Total Transit Time (days)	TMI ΔV (km/s)	MOC ΔV (km/s)	TEI ΔV (km/s)	Total ΔV (km/s)
Cargo	2007	343.2	343.2	3.882	0.831	NA	4.713
Piloted	2009	868.9	290	5.165	3.649	3.772	12.586
Piloted	2009	880.4	350	4.431	2.188	2.601	9.220
Piloted	2009	890.1	400	4.114	1.449	2.044	7.607
Piloted	2009	899.2	450	4.075	1.076	1.536	6.687
Piloted	2009	899.2	450	4.075	2.252†	2.719†	9.226†

Note:
 ΔV s based on 407 km circular orbit at Earth and 250 X 33840 km elliptical Mars parking orbit.
 TMI ΔV includes 100 m/s for plane change
 TEI ΔV includes 150 m/s for apsidal alignment

† 500 km circular Mars parking orbit.

is also utilized on the Mars piloted vehicle's trans-Mars injection (TMI) "drop tanks", primarily to minimize LH₂ boiloff in LEO during the piloted vehicle's 6 month to 1 year long assembly period. Longer duration thermal protection is unnecessary since the TMI drop tanks will be drained during the TMI maneuver and jettisoned immediately afterwards. The installed density of the 2" MLI system is ~2.62 kg/m², and the resulting boiloff rate is ~0.77 kg/m²/month (based on an estimated heat flux of ~0.129 W/m²).

For the "reference" 880 day piloted Mars mission baselined in this study, a 4" MLI system with vapor-cooled shield (VCS) [located midway through the MLI to reduce heat leaks] is assumed. The combined 4" MLI/VCS thermal protection system is used exclusively on the "core stage" and "in-line" tanks which contain LH₂ propellant for periods of time ranging from ~1.5 to 2.5 years. The installed densities of the 4" MLI and VCS systems are ~4.92 kg/m² and 1.26 kg/m², respectively. The resulting boiloff rate is ~0.38kg/m²/month (based on an estimated heat flux of ~0.063 W/m²). Finally, one 0.5 mm sheet of aluminum (corresponding to ~1.35 kg/m²) is included on all LH₂ tanks for micrometeoroid protection.

Modular Lunar/Mars Vehicle Description

In determining the characteristics for a common modular STS, a variety of lunar and Mars mission scenarios were examined. Consideration was also given to component compatibility (in terms of payload mass and volume limits) with a range of HLLV options. A Saturn V-derived HLLV with two LOX/RP boosters was selected as the reference launch system for this study. It is capable of delivering net payloads of 254 t or 230 t to altitudes of 100 nautical miles (185 km) or 220 nautical miles (407 km), respectively. The implications of a reduced HLLV capability (on the order of 150 t to 407 km) was also considered in component selection and is discussed briefly in this study.

First Lunar Outpost TLI Stage

As the size of payloads delivered to the lunar surface increase, the benefits of a NTR lunar transfer stage become more apparent. A sizing analysis was performed during the "Fast Track" Study⁷ to determine attractive NTR engine/stage configurations for the FLO mission. Figure 9, taken from that recently completed study, shows the

UC₂ Particles in Graphite with ZrH Moderator Augmentation
1.2 MWh per Fuel Element, T_c=2550 K, Isp=870 sec

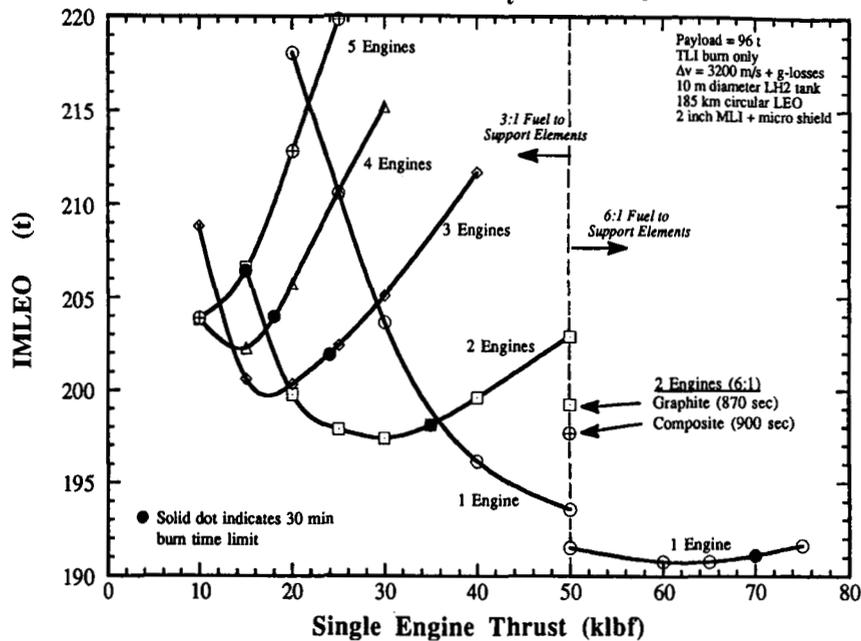


Fig. 9. "First Lunar Outpost", IMLEO Sensitivity to Single Engine Thrust Level

IMLEO required to deliver 96 t (the mass of the piloted FLO lander and its TLI stage adapter) to TLI conditions, as a function of engine thrust level for single and multi-engine stage designs. Figure 9 shows that for a given "total" thrust level, multiple engine configurations have a higher IMLEO. This is due in part to the buildup of inert weight from multiple engine components (e.g., pumps, lines and valves, shielding, etc.) in a "clustered" configuration, and also to the decrease in the engine thrust-to-weight ratio for lower thrust NTR systems (see Table 1). For example, assuming a total thrust of 75 klbf, the three 25 klbf engine configuration has a larger IMLEO requirement (at 202.4 t) than the single 75 klbf vehicle (at 192.9 t).

To prevent the TLI burn times from becoming excessive and to provide margin for the remaining engine(s) in case of an "engine out" occurrence, a "30 minute limit" on burn time (represented by the solid dot on each curve) was specified. Points to the right of the solid dot have burn times less than 30 minutes. Although a single 75 klbf engine stage design has the best performance in terms of IMLEO, a two engine configuration using 50 klbf NTRs has

been chosen as the reference system (both in the Fast Track and in this study) because of its "engine out" capability and the attractiveness of clustered 50 klbf engine configurations for Mars cargo and piloted missions. A large experience base also exists on 50 klbf-class engines (the KIWI-B and NRX reactor series) from the earlier Rover/NERVA programs. Although the Fast Track Study used UC₂ particles in graphite as its reference fuel form, the same characteristics and trends are observed for the composite fuel. A data point showing the reference composite fuel system has been added to Fig. 9 for comparison. Its TLI burn duration is just under 21 minutes.

Figure 10 compares the IMLEO requirements for FLO using NTR and chemical propulsion TLI stages. All of the NTR stages considered have lower IMLEO than the current chemical reference system which uses a single J-2S engine producing ~265 klbf of thrust. A clustered engine configuration using five RL10 A-4 engines is also indicated for comparison. Figure 10 shows quite dramatically that NTR propulsion can enhance the performance capability for the FLO mission.

UC₂ Particles in Graphite with ZrH Moderator Augmentation
1.2 MWth per Fuel Element, T_c=2550 K, Isp=870 sec

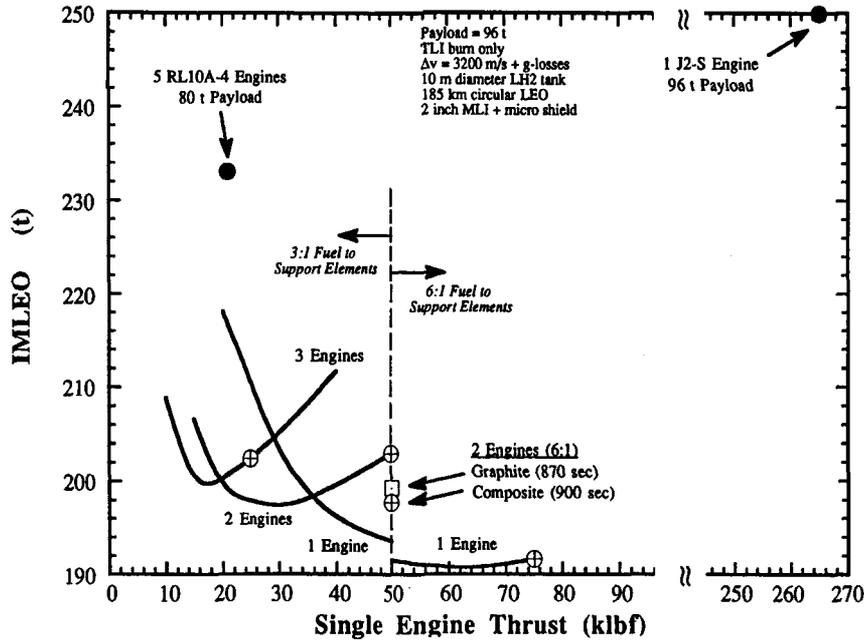


Fig. 10. Benefits of NTR Propulsion for "Fast Lunar Outpost"

The reference NTR stage for FLO and its mass properties are shown in Fig.11. The main LH₂ propellant tank has a 10 m diameter, 14.5 m length and $\sqrt{2}/2$ ellipsoidal domes. The tank is constructed of 2219-T87 Al, has a LH₂ propellant capacity of ~66 t (with an assumed 2.5% ullage), and is designed to handle "4g" launch loads under fully loaded conditions. Avionics, power and RCS are located in the stage forward adaptor section. During launch, loads from the lander and TLI stage are transferred to the HLLV through a cylindrical ring or skirt located at the aft end of the tank. In-space thrust loads from the two 50 klbf NTRs are transferred to the vehicle through the rear conical adaptor or "thrust structure". An external radiation disk shield is assumed on each engine at present. Because of the substantial quantities of cryogenic and storable propellant between the crew and engines, it may be possible to reduce or even eliminate the need for external shielding. Analysis is presently on-going with the DOE to determine actual shielding requirements on the FLO stage.

In addition to the baseline piloted lander, which assumes storable propellant for lunar

ascent and Earth return, the impact on the NTR stage design of using an "all cryogenic" piloted lander (weighing ~76 t)²⁴ was also considered. By extending the tank length to 20 m, a single launch, reusable "2 Burn" mission scenario is possible. Following the TLI burn, this "stretched" NTR stage would target for a "leading edge" encounter with the Moon to set up a "free return" trajectory to Earth. Nearing Earth, the stage would perform a second Earth orbit capture (EOC) burn at high altitude and use its "cooldown thrust" to achieve a desired final parking orbit. The IMLEO required for the reusable "2 Burn" TLI/EOC configuration is ~202.5 t.

With a 150t HLLV capability, a dual launch, Earth orbit rendezvous and dock scenario can be utilized to assemble a two tank vehicle configuration. This approach is capable of delivering into lunar polar orbit an "all cryo" piloted lander weighing ~60 t. The first launch would carry the "core stage" consisting of a 10 m diameter by 20 m long propellant tank containing ~96 t of LH₂, and two 50 klbf composite fuel NTRs. The second launch would carry the piloted lander and a 10 m diameter by 14.5 m long supplemental "in-line" propellant

tank containing ~ 66 t of LH₂. After lander separation and descent, the "in-line" tank would be jettisoned and the "core stage" returned to Earth orbit for reuse. The total IMLEO would be less than 275 t. Although the reuse options mentioned above have the potential to reduce "life cycle" costs, their use also necessitates the development of additional support infrastructure such as a "propellant tanker" or "fuel depot."

Mars Cargo Vehicle

By extending the length of the FLO NTR stage (to ~20 m), upgrading avionics, and increasing fuel cell reactants and RCS propellants, a single launch Mars cargo vehicle is possible. In the cargo mission scenario, a single TMI burn lasting ~24.5 minutes is used for Earth departure. On reaching Mars, the cargo vehicle performs a second 3.5 minute Mars orbit capture (MOC) burn to achieve a 250 x 33,840 km (~24 hour) elliptical parking orbit. At the appropriate time, the Mars cargo lander performs a de-orbit maneuver and uses a combination of aerobraking, parachutes and terminal descent propulsion to land ~50 t of payload on the Mars

surface. After lander separation, the cargo vehicle would circularize at a high Mars orbital altitude for storage and future possible use, or for permanent disposal.

The overall configuration and mass properties for the Mars cargo vehicle are shown in Fig. 12. With the exceptions of the scaled-up avionics, power and RCS propellant requirements, the Mars cargo vehicle is identical to the reusable "2-Burn" TLI/EOC lunar transfer vehicle discussed above and includes many of the same features as the expendable FLO stage (in terms of thermal and micrometeoroid protection, RCS propulsion system hardware, etc.). Two noticeable differences in the cargo vehicle are the absence of the biological external disk shields and the extended cylindrical forward adaptor required to house the increased fuel cell reactants and RCS propellant tanks. The IMLEO is just under 201 t and the overall vehicle height is ~43.2 m. The length available for the Mars cargo vehicle is ~44.8 m. It is set by the length of the Saturn V-derived HLLV's first and second stages (~80.2 m), and the height of the Vertical Assembly Building doors (~125 m).

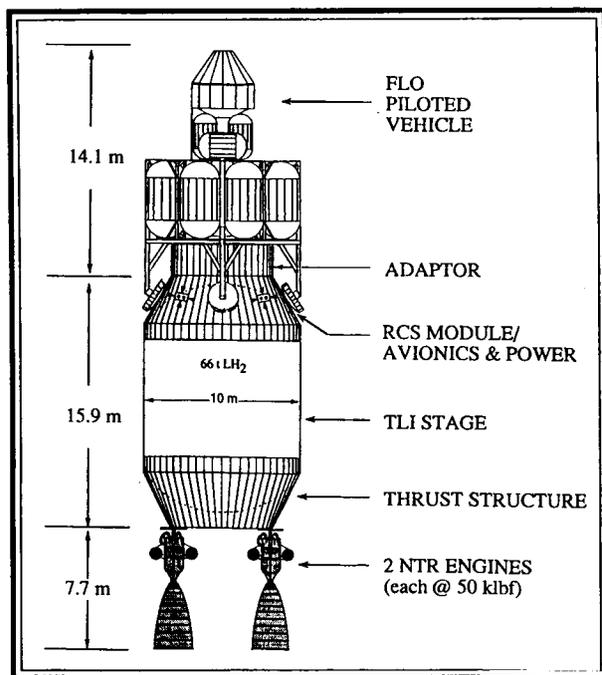


Fig. 11. Vehicle Configuration and Mass Properties for FLO

Expendable FLO TLI Vehicle

Element	Mass (t)
• TLI Stage	13.30
• Avionics and Power	1.00
• Reaction Control	0.46
• NTR Assemblies	
Engines (2)	10.47
External Shields (2)	6.00
• Contingency	3.95
• <u>Dry Mass</u>	<u>35.17</u>
• LH ₂ Propellant	65.48
• RCS Propellant	1.06
• <u>Stage Mass</u>	<u>101.73</u>
• FLO Piloted Vehicle	93.00
• FLO/Stage Adaptor	3.00
• <u>IMLEO</u>	<u>197.73</u>

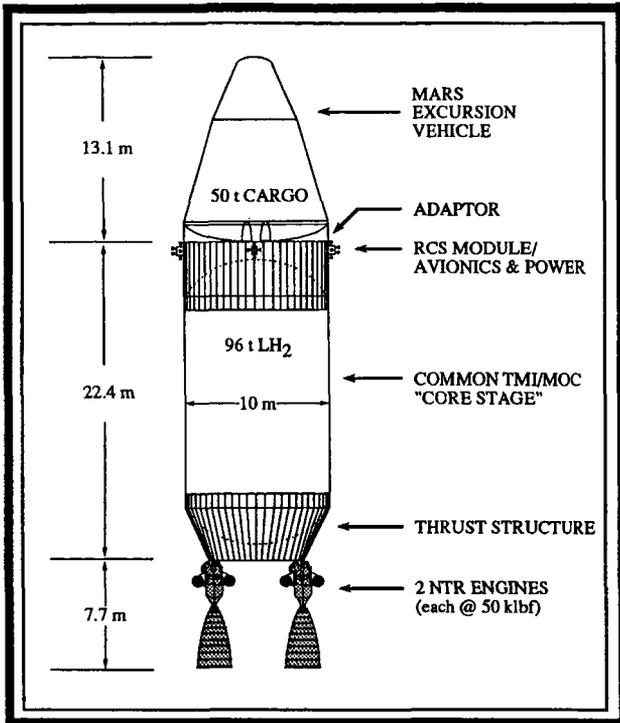


Fig. 12. Mars Cargo Vehicle and Mass Properties

2007 Mars Cargo Vehicle

Element	Mass (t)
• Common TMI/MOC "Core Stage"	18.81
• Avionics and Power	2.00
• Reaction Control	0.48
• NTR Assemblies	
Engines (2)	10.47
External Shields	-
• Contingency	3.70
• <u>Dry Mass</u>	<u>35.46</u>
• LH ₂ Propellant	95.33
• RCS Propellant	5.35
• <u>Stage Mass</u>	<u>136.14</u>
• Mars Excursion Vehicle	63.00
• MEV/Stage Adaptor	1.70
• <u>IMLEO</u>	<u>200.85</u>
Total IMLEO (3 vehicles)	<u>602.55</u>

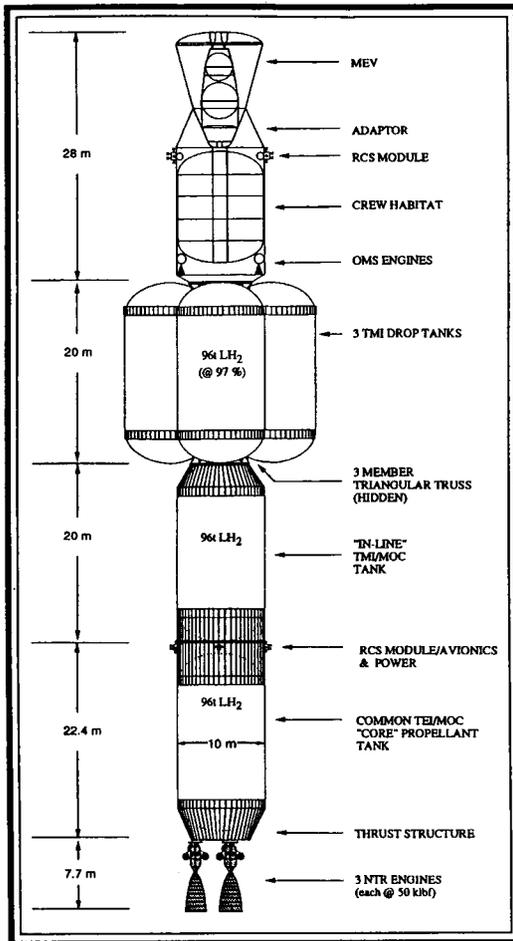
Mars Piloted Vehicle

The 2010 Mars landing mission presently being considered by ExPO is one of the most demanding mission opportunities over the 15 year synodic cycle. Preliminary estimates by the NPO indicate IMLEO requirements in excess of 900 t for a 300 day total transit time/880 day total mission time "fast-conjunction" class mission even with a reduced ΔV 24 hour, elliptical Mars parking orbit. Choosing such a difficult mission as a basis for sizing the components of a Mars STS can result in total thrust levels and propellant tank sizes in excess of those required for the majority of other Mars opportunities, as well as various lunar missions. Perhaps a more fundamental issue is the feasibility of constructing a spacecraft on the order of 1000 t.

For the present study a total mission transit time of 350 days and an elliptical Mars parking orbit were chosen as references (see Table 4). Engine and total thrust levels ranging from 25 to 125 kbf, and from 100 to 250 kbf, respectively, were also examined. The optimum total thrust level was found to be ~150 kbf with two 75 kbf-class

engines providing the lowest IMLEO. Three 50 kbf-class engines were chosen as the reference configuration, however, because of the commonality with the FLO lunar transfer stage and the Mars cargo vehicle (both of which use 50 kbf-class engines). The three engine configuration also allows for the possibility of successful mission completion even with the loss of one engine, an option that does not exist with two engines.

Figure 13 shows the overall configuration and mass properties for the 2010 Mars piloted vehicle. The vehicle consists of a "core stage", an "in-line" LH₂ propellant tank, and three TMI "drop" tanks attached to a preintegrated truss/LH₂ feed system connecting the basic spacecraft to the crew habitat and Mars excursion vehicle. The truss also provides increased separation distance between the habitat and engines further reducing the crew radiation exposure. The standard 10m diameter by 20 m long LH₂ tank developed for the Mars cargo vehicle is used for all five of the propellant tanks found on the piloted vehicle. Differences between tanks lie primarily with the "core stage" and "in-line" tanks which carry a heavier thermal protection system.



2010 Mars Piloted Vehicle

Element	Mass (t)
• MEV (includes ECCV)	75.00
• MEV / Habitat Adaptor	3.09
• Crew Habitat System	59.10
• <u>Total Payload</u>	<u>137.19</u>
• TMI Drop Tanks (3)	47.20
• Preintegrated Truss / LH ₂ Feed System	1.40
• Common TMI / MOC "In-line" Tank	22.70
• Common TEI / MOC "Core Stage"	22.09
• Stage Avionics & Power	2.00
• Reaction Control System	1.04
• NTR Assemblies	
Engines (3)	15.71
External Shields (3)	9.00
• Contingency	13.35
• <u>Vehicle Dry Mass</u>	<u>134.49</u>
• LH ₂ Propellant	471.33
• RCS Propellant	15.45
• <u>IMLEO</u>	<u>758.46</u>

Fig. 13. Mars Piloted Vehicle and Mass Properties

The piloted vehicle is assembled at a 407 km LEO altitude over a 10 month period using five 230 t-class HLLVs with 60 day launch centers. Autonomous rendezvous and docking is assumed between the "core stage", the "in-line" propellant tank and truss structure, and the habitat/MEV components. A robotic arm/manipulator connected to habitat would be used to grapple, position and attach the TMI drop tanks to the truss structure. Following vehicle checkout and "tank top-off" using a small propellant tanker, the piloted vehicle would begin the TMI maneuver. A "3 perigee burn" Earth departure scenario is used which reduces gravity losses to a tolerable 210 m/s. The three drop tanks provide 89% of the usable propellant required for TMI. After they are drained of their propellant and an Earth escape velocity has been achieved, the "spent" drop tanks can be jettisoned to reduce IMLEO further. The remaining 11% of the TMI propellant load is provided by the common "in-line" TMI/MOC

propellant tank. The TMI maneuver requires a total burn time by the three 50 kbf NTRs of ~68 minutes.

After an outbound transfer time of ~191 days, the piloted vehicle initiates the MOC burn which lasts for ~18 minutes. Propellant for this maneuver is provided by both the "in-line" and "core stage" tanks which provide 62% and 38% of the total propellant requirements, respectively. Once in Mars orbit the crew transfers to the MEV and descends to the surface. The 75 t piloted lander is heavier than the Mars cargo lander. It carries 11.5 t of surface payload and the necessary propellant for a 5.264 km/s ascent maneuver. Following a 530 day stay at Mars, the crew returns to the orbiting piloted vehicle in the ascent portion of the MEV. The MEV crew cab is retained for later Earth entry and the remainder of the MEV ascent stage is jettisoned before Mars departure. The "core stage" provides the

propellant for the final TEI burn which lasts ~11.6 minutes. The option also exists to leave Mars with a shortened, less massive vehicle configuration (consisting of the "core stage" and crew habitat system) by undocking and jettisoning the spent "in-line" tank and truss assembly⁵. Following an appropriate cooldown period, the "self-contained" habitat separates from the rest of the spacecraft which is targeted for disposal in heliocentric space. The habitat continues on to Earth using its own power, avionics and auxiliary propulsion for mid-course correction and final Earth targeting maneuvers.

Other Configuration Options

By extending the total in-space transit times from 350 to 450 days, smaller, less massive piloted vehicles can be configured or lower altitude circular Mars parking orbit can be accommodated. Figure 14

shows the variation in IMLEO and vehicle configuration as a function of total mission transit time. By increasing the total transit time to 400 days, the piloted vehicle mass is decreased by 160 t thus reducing to four the number of heavy lift launches required for vehicle assembly. For the 400 and 450 day total transit time missions, the smaller 10 m diameter by 14.5 m long propellant tank developed for FLO would also be utilized in configuring the vehicle. The option of having this second "standardized" tank size provides greater flexibility to the mission planner/vehicle designer, and is one of the key benefits of the modular approach described earlier in Fig. 1. Increasing the total mission transit time to 450 days can also enable a 500 km circular Mars parking orbit to be achieved using the same piloted Mars vehicle shown in Fig. 13. For the "500 km orbit" mission, the three TMI drop tanks would only be filled to 80% of their maximum capacity resulting in an IMLEO of ~687 t.

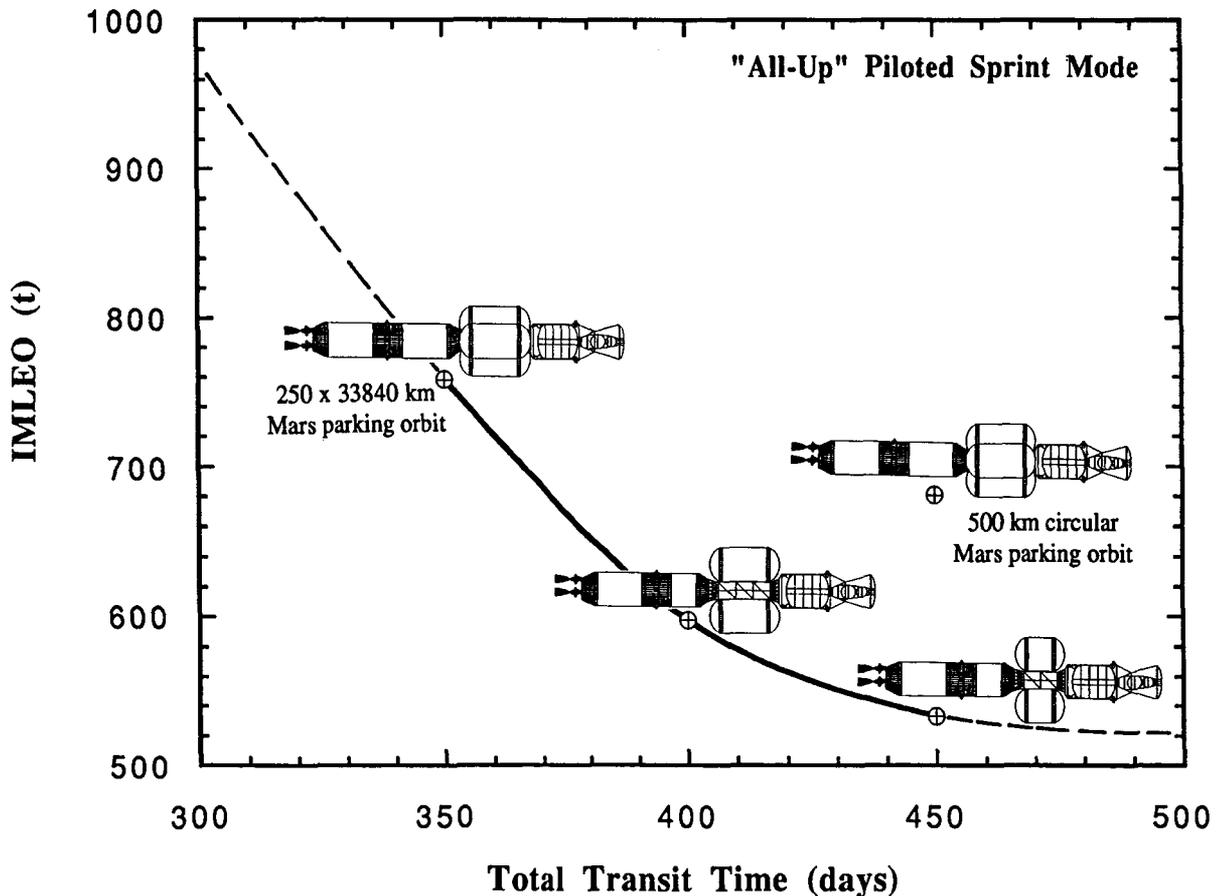


Fig. 14. "2010 Mars Piloted Mission", IMLEO Sensitivity to Total Transit Time and Mars Parking Orbit

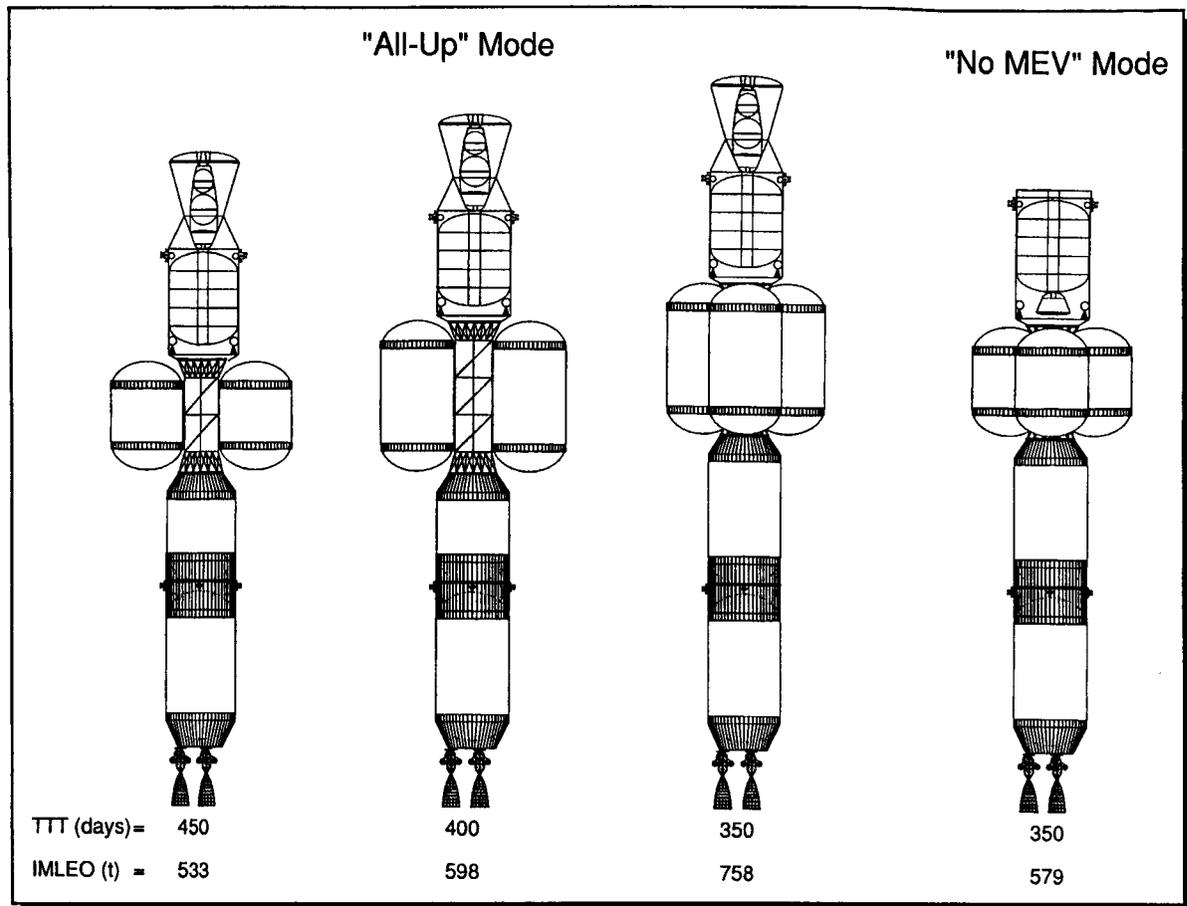


Fig. 15. 2010 Mars Piloted Vehicle Configurational Options

Figure 15 shows, in greater detail, possible configurational options for the 2010 Mars piloted mission as a function of total transit time and mission mode. With the "No MEV" mission mode, the IMLEO required for the 350 day transit time mission can be reduced by ~180 t. Eventually, with the establishment of permanent Mars outpost, refueling of the Mars piloted vehicle, using either a propellant tanker or "in-situ" Martian propellant (the "No MEV/No TEI propellant" mode) may become more acceptable. Under these circumstances, piloted vehicles with IMLEO's on the order of 400 to 500 t can be realized. Figure 16 summarizes the key components of the modular NTR approach.

NTR System Development Program

The NPO, in conjunction with the DOE, has

formulated an overall program for NTR system development which features facility and technology development (both nuclear and non-nuclear), system definition and preliminary design, and "concurrent" advanced development/flight qualification programs. Major milestones of the program are shown in Fig. 17 and include (1) the authority to proceed (ATP) with advanced development and flight qualifications in 1996, and (2) the demonstration of a technology readiness level justifying flight qualification (TRL-6) through full system ground testing of "flight-type" hardware in 2003. These two important milestones would be followed by (3) initial lunar cargo and piloted flights in 2005 and 2006; (4) Mars cargo flights in 2007; and (5) an initial Mars piloted flight in 2009.

Basic "Building Blocks" of Modular NTR Approach

- 50 kbf Engines (used in clusters of 2 or 3)
- 2 "Standardized" Tank Sizes
- Pre-integrated Truss/LH₂ Feed System

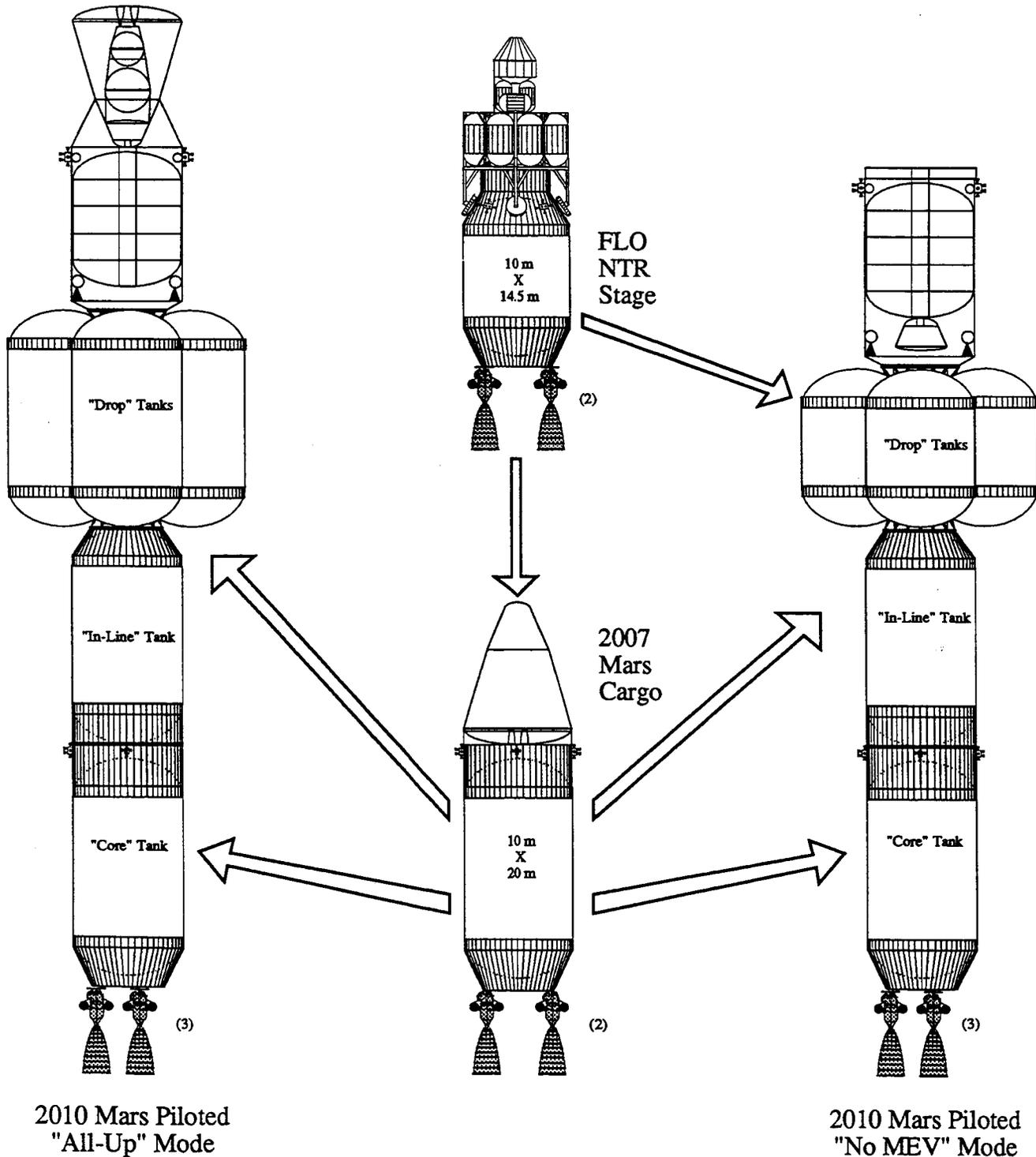


Fig. 16. Key Components of a Modular, NTR-Based Lunar/Mars Space Transportation System

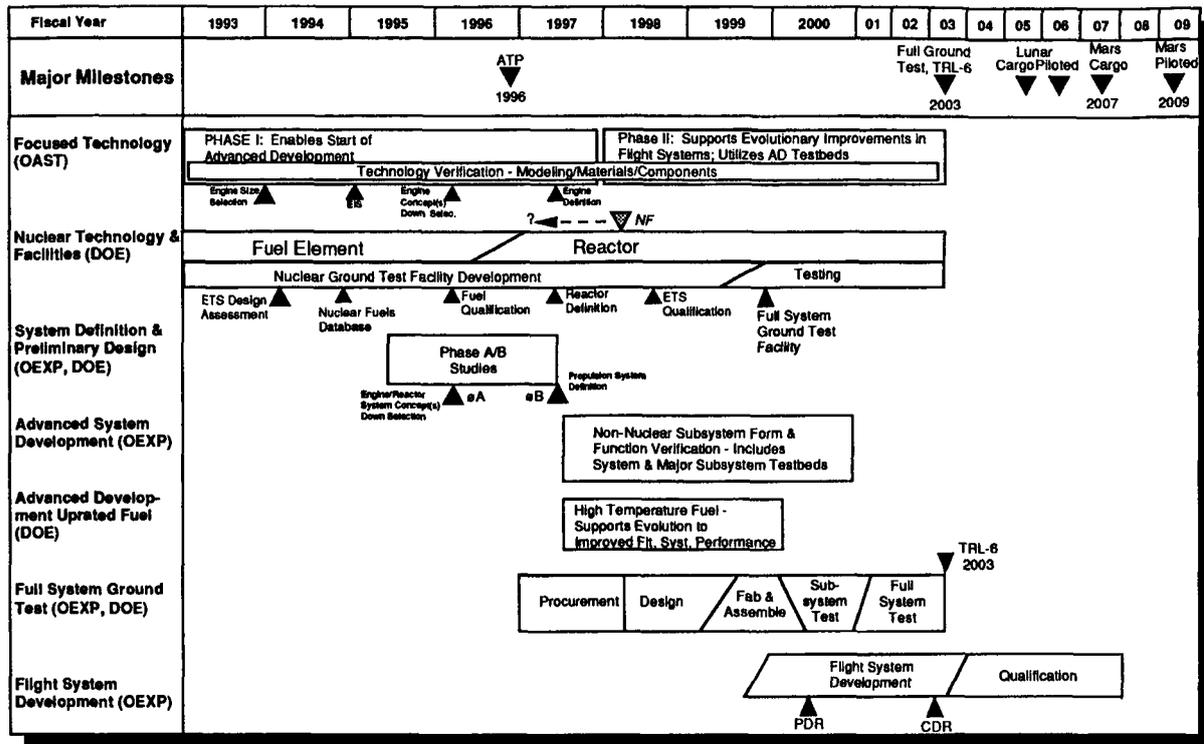


Fig. 17. Overall NTR Propulsion System Development Plan

Technology and Facility Development

Focused technology development is funded by NASA's Office of Aeronautics and Space Technology and includes those activities required to provide the non-nuclear technology data and systems studies information necessary for a positive ATP decision. In Phase I of this activity, a reference engine concept will be defined at a "target" technology level (e.g. hydrogen exhaust temperature = 2700 K, Isp = 900 s), against which other candidate NTR concepts will be compared. Engine size selection will be based primarily on total life cycle costs for the entire exploration program, and on astronaut safety and system reliability issues. The reference engine system will also be designed to accommodate upgrades as improved technologies are brought "on-line". An Environmental Impact Statement (EIS) for the engine development project will also be prepared early in the project.

A down-select to one or two engine concepts will occur when fuel qualification tests, and Phase A

system studies are complete. Phase B system studies of the selected concept(s) will then be completed providing the reactor, engine, and propulsion system design and definition necessary to support the ATP. Only one concept will be selected for advanced development, full system ground test, and flight system development to minimize development cost. Phase II of the focused technology development effort provides for improvements in both the system design and technology used in the reference system to ensure optimum safety, reliability and performance of the resulting propulsion system.

Nuclear technology development will focus initially on fuels production, properties evaluations, and the establishment of a consistent temperature, life, and fission product release data base, using existing reactor facilities. Nuclear testing will be conducted at DOE test sites, and nuclear test facilities have been identified as critical path, high cost development activities. An early critical study of effluent treatment system (ETS) options is

included, and prototype ETS systems must be tested and qualified, leading to design and construction of the full system ground test facility. Because initial facility design and site selection activities must begin before a final engine concept selection is made, a versatile design will be required. Appropriate safety studies, facility environmental impact statements and site selection processes are also included as appropriate.

Development Plans

Phase A studies of multiple concepts are planned in 1995. These are followed by Phase B preliminary design studies of one or two concepts to support the ATP decision. Advanced development component, and major subsystem tests will be conducted in test-beds to verify form and function, including tests to failure to establish appropriate safety and reliability margins. Finally, flight systems hardware will be designed and ground tested verifying technology readiness. Preliminary Design Reviews (PDR) and Critical Design Reviews (CDR) are planned in parallel with the ground testing to enable the early flights.

Cost Analysis

The cost estimates for the multiple lunar and Mars systems described above build on earlier estimates for a lunar NTR stage performed during the Fast Track Study.⁷ The estimates are broken down for the different classes of missions considered, and include two lunar flights, three Mars cargo missions and an initial Mars piloted mission. The analysis approach used consisted of contractor estimates, DOE estimates, grass-roots estimates and parametric cost models such as the Marshall Space Flight Center Launch Vehicle Cost Model for the stage subsystems.

Assumptions and Ground Rules

All costs were estimated in 1992 dollars, phased and converted to real year dollars using the NASA R&D Inflation Index according to the following milestones: (1) program start in 2000; (2) a full engine ground test in 2003; (3) a first lunar cargo flight in 2005; (4) a first Mars cargo flight in 2007; and (5) an initial Mars piloted flight in 2009. The cost estimates included flight system development, testing and fabrication of the 50 kbf NERVA-derived engines and stage components required for the two lunar and four Mars vehicles. Specifically

excluded from these estimates are the costs associated with launch integration and operations, mission operation, on-orbit assembly of the Mars piloted vehicle, and the costs of the LH₂ propellant.

Three engines and one stage were costed for system test hardware. The engines were used for cold flow testing, full power reactor development testing, and engine qualification. Additional test hardware (and the number of units involved) included the reactor (2), fuel elements (135), nozzle (3.2) and nozzle skirt (1.5), and turbopumps (4.3). In addition to the above test hardware, one complete set of engine and stage components were costed as spares for the program. For each new NTR system application, an additional set of components was included for spares or additional test hardware.

A prime contractor was assumed for both the engine and stage development efforts. In the engine area, a reactor subcontractor was also assumed and contractor fees reflecting this relationship are included. Prime contractor "wraps" which include integration and checkout, test operations, system engineering and program management have been applied for launch vehicles and engines. Contractor estimates were used for ground support and tooling and a contractor fee of 10% was included.

Total program reserve was estimated at 35% of the prime contractor costs while the engine and stage reserves were estimated at 40% and 30%, respectively, using the NASA Headquarters Risk/Reserve Model. Both the engine and stage provide approximately the same contribution to the total cost. Finally, government program support was included at 15%.

NASA/DOE Testing and Support

An assessment of facilities required for a NTR and stage development effort was conducted by NASA and has indicated the need for propellant tank, nozzle, turbopump, and feed system test facilities. Also identified were facilities for instrumentation and control simulation, acoustic testing, and engine dynamic testing. The costs to modify and/or construct the above facilities was estimated to be ~\$60 million (in '92\$).

Information on nuclear component/system testing and support was provided by the Los Alamos National Laboratory (LANL) and the Idaho National

Engineering Laboratory. In a recently completed study,²⁵ LANL assessed the suitability of the Jackass Flats nuclear facilities, developed during the Rover/NERVA programs, for current day NTR testing. The study examined the facility requirements for testing 25 to 100 kbf-class engines assuming burn times on the order of an hour or more. The LANL estimate of total cost for an engine system test site using the Rover/NERVA facilities was about \$253 million (in '92\$). This estimate included refurbishment costs for the E-MAD assembly and disassembly facility, the ETS-1 engine test stand, the R-MAD storage facility, and the interconnecting railroad for intra-site transportation. Also included in the ETS-1 test stand estimate was the cost of an effluent treatment system to prevent radiological releases to the environment.

Using the LANL study results and those obtained in the earlier Fast Track assessment, nuclear facilities construction and operation costs were derived. Additional cost estimates were provided by LANL for fuel fabrication facility modifications, and an electric furnace test facility. DOE costs were also provided for diagnostic support and program management.

The DOE costs are primarily related to development and include costs associated with reactor/engine testing leading up to the successful completion of the qualification engine test program

for the first lunar or Mars flight. The only recurring costs are for testing and program management of nuclear component production, and for fuel procurement and fuel element fabrication. Fuel costs were estimated at \$25 thousand per kilogram of enriched uranium.

Cost Summary

In order to quantify the cost benefits of the modular, lunar/Mars vehicle approach, three different mission implementation scenarios were examined. In Scenario I, a combined cryogenic/storable lander and chemical TLI stage is used for the First Lunar Outpost, and the NTR system is introduced on the Mars cargo mission. In Scenario II, the chemical TLI stage is replaced by a NTR stage and a "scaled up" TLI stage is used for the Mars cargo mission. Finally, in Scenario III, the use of an "all cryo" lander enables a reusable lunar NTR stage suitable for use on the Mars cargo and piloted missions. Table 5 summarizes the development (DDT&E), flight hardware, and total missions costs for these three scenarios. Scenario III develops a single propellant tank (10 m in diameter and 20 m long) which is used in all subsequent lunar and Mars missions. It has the lowest total mission cost. Scenario II develops two tank sizes -- a 14.5 long tank used in the expendable TLI stage for FLO, and a "stretched" 20 m long tank used for the Mars cargo mission. Because of the increased flexibility in vehicle design provided by two standard tank sizes,

Table 5. Mission Cost Benefits of Early NTR Utilization

Mission		Lunar/Mars Mission Costs (92M\$)		
		Scenario I	Scenario II	Scenario III
Lunar Outposts (2 Flights)	DDT&E	2316.3	3329.5	3379.6
	Flight Hardware	383.4	615.1	648.6
Mars Cargo (3 Flights)	DDT&E	3272.3	1014.3	825.2
	Flight Hardware	1243.6	1043.6	1043.6
Mars Piloted (1 Flight)	DDT&E	1226.4	1226.4	1226.4
	Flight Hardware	901.8	901.8	901.8
Total Missions Cost	DDT&E	6815.0	5570.2	5431.2
	Flight Hardware	2528.8	2560.5	2594.0
	DOE	1080.3	1188.8	1188.8
Total	92 M\$	10424.1	9319.5	9214.0
Total	RY M\$	18747.6	16766.6	16551.5

and the small cost difference (~\$106 million in '92\$), Scenario II is favored at this time.

The cost for the NTR systems reflect the modular design approach with the first application of the stage or subsystem receiving all the development and non-recurring facilities costs. Costs for subsequent uses reflect only development cost deltas and the additional flight hardware. Table 5 shows that savings of over \$1 billion (in '92\$) may be achieved if a NTR lunar transfer stage is developed to support the FLO mission. This savings results primarily from elimination of the chemical TLI stage development effort. Real year (RY) dollar savings can be even greater (~\$2 billion) if scarce NASA resources can be redirected away from "limited use" technologies and into a focused NTR development effort early in the program. Figure 17 shows graphically a comparison of the cumulative costs for the implementation scenarios discussed here.

Summary and Conclusions

The rationale and benefits of developing a common, modular lunar/Mars space transportation system, based on NTR propulsion, is presented. Key components of the modular NTR approach are described and consist of (1) a 50 klbf NERVA-derived engine used in clusters of 2 or 3; (2) two "standardized" tank sizes developed for the First Lunar Outpost and Mars cargo vehicle applications; and (3) a preintegrated truss/propellant feed system used for transferring LH₂ from the TMI drop tanks into the "in-line" tank. By using these components in a "building block" fashion, a variety of single and multi-engine lunar and Mars vehicles can be configured to satisfy particular mission requirements.

For NASA's FLO mission, an expendable NTR stage powered by two 50 klbf engines is capable of delivering the 93 t FLO lander and its 3 t adaptor to

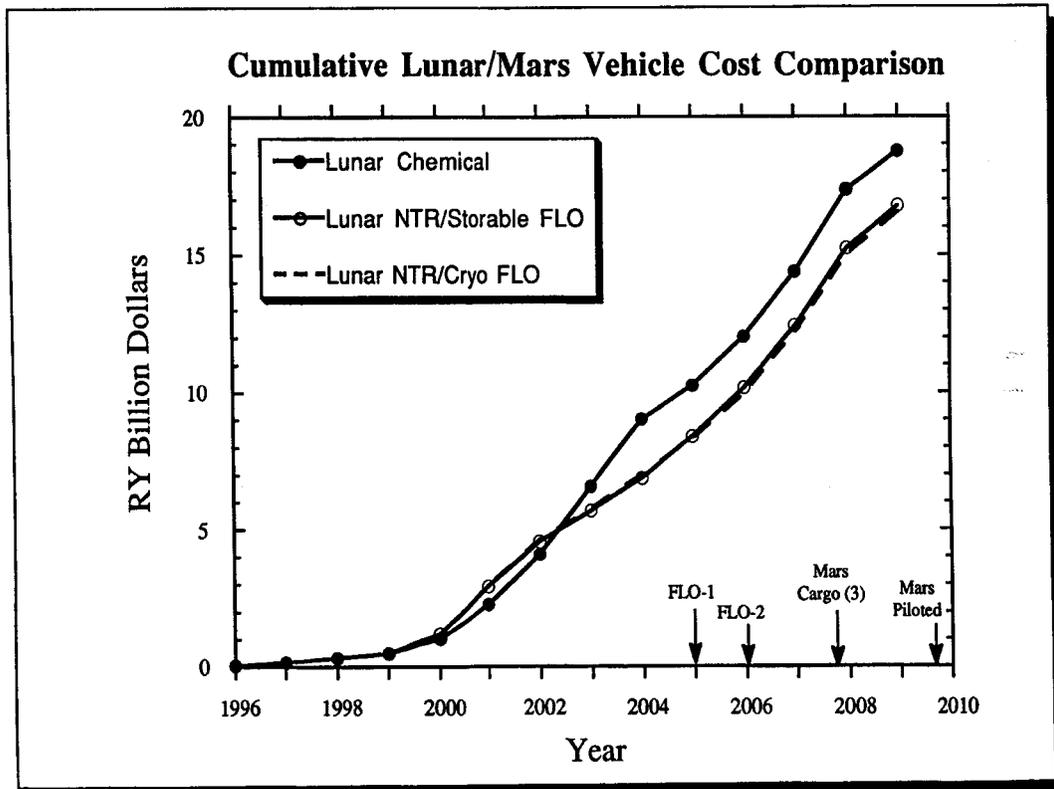


Fig. 18. Cumulative Lunar/Mars Vehicle Cost Comparison

TLI conditions for an IMLEO of ~198 t compared to 250 t for a LOX/LH₂ chemical stage. By extending the stage LH₂ tank length (from ~14.5 m to 20 m) and capacity (from ~66 t to 96 t), a single launch Mars cargo vehicle capable of delivering ~50 t of surface payload is possible. With an "all cryogenic" FLO lander as its payload, this same "stretched" FLO stage would form the basis for a reusable, "2-burn" TLI/EOC lunar stage. The three "building blocks" listed above would be used to configure a variety of Mars piloted vehicles depending on total mission transit time and/or Mars parking orbit requirements.

The programmatic supporting NTR system development is presented along with individual and cumulative cost estimates for the lunar and Mars

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13. ABSTRACT (Maximum 200 words) The results of integrated systems and mission studies are presented which quantify the benefits and rationale for developing a common, modular lunar/Mars space transportation system (STS) based on nuclear thermal rocket (NTR) technology. At present, NASA's Exploration Program Office (ExPO) is considering chemical propulsion for an "early return to the Moon," and NTR propulsion for the more demanding Mars missions to follow. The time and cost to develop these multiple systems are expected to be significant. The Nuclear Propulsion Office (NPO) has examined a variety of lunar and Mars missions and heavy lift launch vehicle (HLLV) options in an effort to determine a "standardized" set of engine and stage components capable of satisfying a wide range of Space Exploration Initiative (SEI) missions. By using these components in a "building block" fashion, a variety of single and multi-engine lunar and Mars vehicles can be configured. For NASA's "First Lunar Outpost" (FLO) mission, an expendable NTR stage powered by two 50 kbf engines can deliver ~96 metric tons (t) to trans-lunar injection (TLI) conditions for an initial mass in low Earth orbit (IMLEO) of ~198 t compared to 250 t for a cryogenic chemical TLI stage. The NTR stage liquid hydrogen (LH ₂) tank has a 10 m diameter, 14.5 m length, and 66 t LH ₂ capacity. The NTR utilizes a UC-ZrC-graphite "composite" fuel with a specific impulse (Isp) capability of ~900 s and an engine thrust-to-weight ratio of ~4.3. By extending the size and LH ₂ capacity of the lunar NTR stage to ~20 m and 96 t, respectively, a single launch Mars cargo vehicle capable of delivering ~50 t of surface payload is possible. Three 50 kbf NTR engines and the two standardized LH ₂ tank sizes developed for lunar and Mars cargo vehicle applications would be used to configure the Mars piloted vehicle for a mission as early as 2010. The paper describes the features of the "common" NTR-based Moon/Mars STS, examines performance sensitivities resulting from different "mission mode" assumptions, and quantifies potential schedule and cost benefits resulting from this modular Moon/Mars NTR vehicle approach.				
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